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K.H. Liew, E. Urip, and S.L. Yang
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A PARAMETRIC CYCLE ANALYSIS OF A SEPARATE-FLOW TURBOFAN WITH INTERSTAGE TURBINE BURNER

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Today's modern aircraft is based on air-breathing jet propulsion systems, which use moving fluids as substances to transform energy carried by the fluids into power. Throughout aero-vehicle evolution, improvements have been made to the engine efficiency and pollutants reduction. This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (*ITB*). The *ITB* considered in this paper is a relatively new concept in modern jet engine propulsion. The *ITB* serves as a secondary combustor and is located between the high- and the low-pressure turbine, i.e., the transition duct. The objective of this study is to use design parameters, such as flight Mach number, compressor pressure ratio, fan pressure ratio, fan bypass ratio, linear relation between high- and low-pressure turbines, and high-pressure turbine inlet temperature to obtain engine performance parameters, such as specific thrust and thrust specific fuel consumption. Results of this study can provide guidance in identifying the performance characteristics of various engine components, which can then be used to develop, analyze, integrate, and optimize the system performance of turbofan engines with an *ITB*.

I. Introduction

In most common air-breathing propulsion engines, fluid air is used as a medium to convert air-fuel mixture into kinetic energy. The energy is used for different applications. These engines convert high pressure and high temperature gas from the combustion chamber into work through the turbine to power the fan for turbofan, shaft for turbo-shaft, and propeller for turboprop, in addition to drive the compressor and accessories. Increases in pressure and momentum across the engine produce sufficient thrust to power the aircraft.

Throughout aero-vehicle evolution, scientists and engineers have attempted to improve engine efficiency, to make it smaller, lighter, require less fuel consumption, and yet more powerful. This type of engine is suitable for combat military aircrafts and long-range commercial aircrafts, since most of their designs are constrained by the weight of the engines and the distance of the flight. Lighter engines mean the aircraft can carry more payload and fuel for long combat operation or long flight time. Scientists have proposed solutions of how to achieve these goals, and one of them is introducing an Interstage Turbine Burner (*ITB*) into the engines.

Almost all commercial aircraft engines have transition duct between the high-pressure (*HPT*) and the low-pressure turbine (*LPT*). The *ITB* considered in this study, which is a relatively new concept in gas turbine engines, is to use the transition duct as a secondary combustor. By doing so, no new component is added to the existing system. One should note that *ITB* being studied here is equivalent to the 1-*ITB* engine in Liu and Sirignano[1], where the 1-*ITB* can be conveniently located between *HPT* and *LPT* or in the stator of the turbine stages.

The major advantages associated with the use of *ITB* are an increase in thrust and reduction in NOx emission, as illustrated in Figure 1. In Figure 1a, the inlet temperature of the *HPT* remains unchanged. As the flow undergoes secondary combustion, a higher specific thrust (*ST*) results. This implies a smaller and lighter engine and hence, lower cost and higher payload. Figure 1b shows the case in which the peak temperature inside the primary combustor is decreased; therefore the amount of thermal NO_x production can be reduced. Furthermore, by lowering the temperature of the primary combustor and the *HPT*, a smaller amount of cooling air is required. Another advantage is the safety improvement, where flameout of either main burner or the *ITB* will not shut down the engine.

Sirignano and Liu[1,2] mention that one major

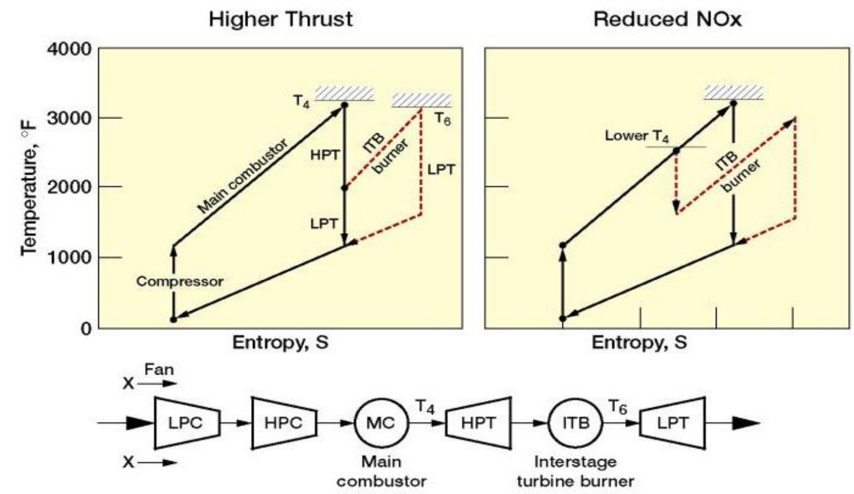


Figure 1 Thermodynamic cycles of a turbofan engine with ITB (a) higher thrust (b) reduced NO_x with lower peak temperature inside the combustor

consequence of increasing engine thrust-to-weight ratio is that, combustion residence time may become shorter than the time required to complete combustion. Therefore, complete combustion process will take place in the turbine passages. This is generally undesirable because it is too technically difficult to burn fuel in a turbine rotor. Constructing what they called *M-ITB*, between turbine stages remedies this problem, where *M* represents a number of *ITBs*. The study in Liu and Sirignano[2] also shows that turbofan base engine and turbofan base engine with afterburner do not compete much with a turbofan base engine with *M-ITB*. Introducing an afterburner to the engine does increase specific thrust but at the expense of the thrust specific fuel consumption (*TSFC*). One expects to see an increase in *ST* and a decrease in *TSFC* in the final design of the engine. Unfortunately, an increase in *ST* always results in an increase of *TSFC*, for constant thermal efficiency. Nevertheless, a turbofan engine with *M-ITB* results in a tremendous increase in *ST* and only small increase in *TSFC*. Turbofan base engines with *ITB* increase its engine performance further by varying engine design parameters, such as compressor pressure ratio (*CPR*), fan pressure ratio (*FPR*), and bypass ratio (*BR*), in addition to the new material that can sustain higher pressure and higher temperature.

This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (*ITB*), which is also known as the *on-design* analysis.

II. Aircraft Engine Performance Parameters

There are several of the air-breathing engine performance parameters that are useful in aircraft propulsion design. The first performance parameter is the thrust of the engine for sustaining flight. Thrust is the force produced due to momentum and pressure increases across the engine. It is used to sustain (thrust = drag), accelerate (thrust > drag), or decelerate (thrust < drag) a flight. To increase thrust, one can introduce an additional nozzle to raise jet velocities or add an afterburner.

ST is defined as thrust (*T*) produced per unit mass airflow rate. In the other words, it defines the amount of fluid air needed to produce a level of thrust. It also means how effective the size of the engine can produce a certain amount of thrust. The specific thrust is described by the following equation:

$$ST = \frac{T}{\dot{m}_{air}} \tag{1}$$

In this paper, $\dot{m}_{air}$ is defined as the air mass flow rate of the engine core, $\dot{m}_c$.

However, one would like to know how significant the thrust increase is compared to the amount of fuel being injected. This leads to the definition of *TSFC*, which is the second performance parameter. The *TSFC* defines the rate of total mass flow rate of fuel per unit thrust produced. Accordingly,

$$TSFC = \frac{\dot{m}_f}{T} \tag{2}$$

This equation describes how effective a mass unit amount of fuel injected can produce thrust. Small *TSFC* indicates small fuel consumption for the same level of thrust produced, which is generally sought.

Other useful engine performance parameter is thermal efficiency, which is defined as the net rate of the kinetic energy gain out of the engine divided by the rate of thermal energy available from the fuel:

$$\eta_{th} = \frac{\dot{E}_{kinetic,gain}}{\dot{m}_f \cdot h_{PR}} \tag{3}$$

III. Parametric Cycle Analysis for A Separate-Flow Turbofan Engine with *ITB*

In this study, the air stream entering turbofan engine (station number 0) will flow through the fan and the engine core separately. The fan increases the propellant mass flow rate with an accompanying decrease in the exit velocity for a given thrust.

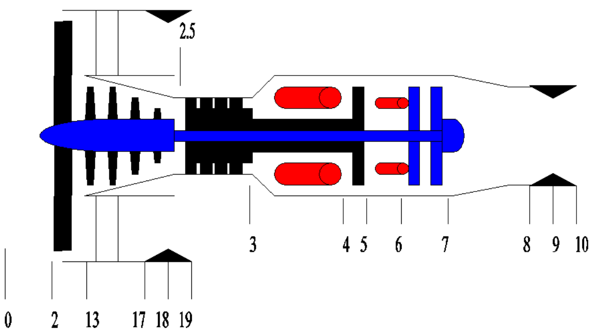


Figure 2 Station numbering of a turbofan engine with *ITB*

The station numbering for the turbofan cycle analysis with *ITB* is given in Figure 2, in which the *ITB* (the transition duct) is located between station 5 and 6. The station numbering and the calculation steps follow closely as described in Mattingly[3]. We assume that (1) the working fluid is air which behaves as a perfect gas with constant properties at three different sections: γ_c, R_c, C_{pc} for gas upstream of main burner, i.e., station 3; γ_b, R_b, C_{pb} for gas between station 4 and 5, i.e., across the high-pressure turbine; γ_t, R_t, C_{pt} for gas downstream of *ITB*, i.e., station 6, where γ is the specific heat ratio, R is the specific gas constant, and C_p is the constant pressure specific heat, (2) all components are adiabatic (i.e. no turbine cooling. However, effects of pressure and work losses due to turbine cooling can be taken into account by using uncooled turbine efficiency reduced by 1 to 3 percent as suggested by Hawthorne[4].); and (3) Constant polytropic efficiencies[3] of compressor, turbine and fan will be used to relate the stage pressure ratio π to their temperature ratio τ. Cycle analysis is then applied to both the bypass stream and engine core stream separately as listed below.

Bypass stream

In this study, we consider only the uninstalled thrust (F), which depends on the engine alone and hence is independent of the nacelle. The uninstalled thrust of bypass stream (F_{fan}) is given by:

$$F_{fan} = \frac{\dot{m}_{fan}}{g_c}(V_{19} - V_0) + A_{19}(P_{19} - P_0) \tag{4}$$

To express equation (4) in term of the free stream Mach number and sound speed (M_0 and a_0), temperature (T) and pressure (P), it can be rearranged to give:

$$\frac{F_{fan}}{\dot{m}_{fan}} = \frac{a_0}{g_c}\left[\frac{V_{19}}{a_0} - M_0 + \left(\frac{T_{19}/T_0}{V_{19}/a_0}\right)\left(\frac{1 - P_0/P_{19}}{\gamma_c}\right)\right] \tag{5}$$

The velocity ratio V_{19}/a_0 can be expressed in term of local Mach number, temperature, pressure, and gas properties as follows:

$$\left(\frac{V_{19}}{a_0}\right)^2 = \frac{a_{19}^2 M_{19}^2}{a_0^2} = \left(\frac{T_{19}}{T_0}\right) M_{19}^2 \tag{6}$$

The ratio of fan exit temperature to ambient temperature T_{19}/T_0 and fan exit Mach number M_{19} are calculated by the following equations:

$$\frac{T_{19}}{T_0} = \frac{T_{t19}/T_0}{T_{t19}/T_{19}} = \frac{T_{t19}/T_0}{\left(P_{t19}/P_{19}\right)^{(\gamma_c - 1)/\gamma_c}} \tag{7}$$

$$M_{19}^2 = \frac{2}{\gamma_c - 1}\left[\left(\frac{P_{t19}}{P_{19}}\right)^{(\gamma_c - 1)/\gamma_c} - 1\right] \tag{8}$$

The total-static temperature (T_t/T_0) and pressure (P_t/P_0) ratio can be calculated as:

$$\frac{T_{t19}}{T_0} = \frac{T_{t0}}{T_0}\frac{T_{t2}}{T_{t0}}\frac{T_{t13}}{T_{t2}}\frac{T_{t19}}{T_{t13}} = \tau_r \tau_d \tau_{fan} \tau_{fn} \tag{9}$$

$$\frac{P_{t19}}{P_{19}} = \frac{P_0}{P_{19}}\frac{P_{t0}}{P_0}\frac{P_{t2}}{P_{t0}}\frac{P_{t13}}{P_{t2}}\frac{P_{t19}}{P_{t13}} = \frac{P_0}{P_{19}}\pi_r \pi_d \pi_{fan} \pi_{fn} \tag{10}$$

where the free stream total/static temperature and pressure ratios (τ_r and π_r) are given by:

$$\tau_r = \frac{T_{t0}}{T_0} = 1 + \frac{\gamma - 1}{2} M_0^2 \tag{11}$$

$$\pi_r = (\tau_r)^{\gamma/(\gamma-1)} \tag{12}$$

Engine core stream

The engine core stream cycle analysis is similar to the bypass stream except it includes energy addition/subtraction across all components in the engine.

The uninstalled thrust of the engine core stream (F_c) is given by:

$$F_c = \frac{1}{g_c}\left(\dot{m}_{10} V_{10} - \dot{m}_c V_0\right) + A_{10}\left(P_{10} - P_0\right) \tag{13}$$

Rearranging and expressing equation (13) in term of free stream Mach number and sound speed, temperature and pressure gives:

$$\frac{F_c}{\dot{m}_c} = \frac{a_0}{g_c}\left[(1+f)\frac{V_{10}}{a_0} - M_0 + (1+f)\frac{R_t}{R_c}\left(\frac{T_{10}/T_0}{V_{10}/a_0}\right)\left(\frac{1 - P_0/P_{10}}{\gamma_c}\right)\right] \tag{14}$$

where the fuel-air ratio f is the sum of the fuel-air ratio in the main combustor (f_b) and in the *ITB* (f_{itb}):

$$f = f_b + f_{itb} \tag{15}$$

The velocity ratio V_{10}/a_0 can be expressed in term of local Mach number, temperature, pressure, and gas properties as:

$$\left(\frac{V_{10}}{a_0}\right)^2 = M_{10}^2 \left(\frac{\gamma_t}{\gamma_c}\right)\left(\frac{R_t}{R_c}\right)\left(\frac{T_{10}}{T_0}\right) \tag{16}$$

where

$$M_{10}^2 = \frac{2}{\gamma_t - 1}\left[\left(\frac{P_{t10}}{P_{10}}\right)^{(\gamma_t - 1)/\gamma_t} - 1\right] \tag{17}$$

$$\frac{T_{10}}{T_0} = \frac{T_{t10}/T_0}{T_{t10}/T_{10}} = \frac{T_{t10}/T_0}{\left(P_{t10}/P_{10}\right)^{(\gamma_t - 1)/\gamma_t}} \tag{18}$$

The total-static temperature and pressure ratio can be calculated as:

$$\frac{T_{t10}}{T_0} = \tau_r \tau_d \tau_{lpc} \tau_{hpc} \tau_b \tau_{hpt} \tau_{itb} \tau_{lpt} \tau_n \tag{19}$$

$$\frac{P_{t10}}{P_{10}} = \frac{P_0}{P_{10}} \pi_r \pi_d \pi_{lpc} \pi_{hpc} \pi_b \pi_{hpt} \pi_{itb} \pi_{lpt} \pi_n \tag{20}$$

Assuming isentropic processes in the inlet (diffuser) and exit (nozzle), it yields

$$\tau_n = \tau_d = 1 \tag{21}$$

Others, such as π_d, π_b, π_{itb}, π_n, and π_{fn}, are input parameters. The compressor pressure ratio (π_c) is the product of *LPC* and *HPC* pressure ratio and is one of the design parameters

$$\pi_c = \pi_{lpc} \pi_{hpc} \tag{22}$$

Main burner (station 3-4)

Application of the steady flow energy equation to the main burner gives

$$\dot{m}_c C_{pc} T_{t3} + \dot{m}_b \eta_b h_{PR-b} = \dot{m}_4 C_{pb} T_{t4} \tag{23}$$

The ratio between total enthalpy of the main burner exit and ambient enthalpy, denoted by $\tau_{\lambda\text{-}b}$, is defined as

$$\tau_{\lambda-b} = \frac{\left(C_p T_t\right)_{burner\ \ exit}}{\left(C_p T_t\right)_{0,\ \ free\ \ stream\ \ (ambient)}} \tag{24}$$

and is an input design parameter. Rearranging and solving equation (23) for f_b yields

$$f_b = \frac{\tau_r \tau_c - \tau_{\lambda-b}}{\tau_{\lambda-b} - \dfrac{\eta_b h_{PR-b}}{C_{pc} T_0}} \tag{25}$$

Interstage Turbine Burner (station 5-6)

Application of the steady flow energy equation to the *ITB* gives

$$\dot{m}_4 C_{pb} T_{t5} + \dot{m}_{itb} \eta_{itb} h_{PR-itb} = \dot{m}_6 C_{pt} T_{t6} \tag{26}$$

Similarly, we introduce $\tau_{\lambda\text{-}itb}$, which is the ratio between the total enthalpy of the *ITB* exit and the ambient enthalpy

$$\tau_{\lambda-itb} = \frac{(C_p T_t)_{itb\ \ exit}}{(C_p T_t)_{0,\ \ free\ \ stream\ \ (ambient)}} \tag{27}$$

Rearranging and solving equation (26) for f_{itb} yields

$$f_{itb} = \frac{\tau_r \tau_c \tau_b \tau_{hpt} - \dfrac{C_{pc}}{C_{pb}} \tau_{\lambda-itb}}{\dfrac{C_{pc}}{C_{pb}} \tau_{\lambda-itb} - \dfrac{\eta_{itb} h_{PR-itb}}{C_{pb} T_0}} (1 + f_b) \tag{28}$$

Applying the energy equation to compressors, turbines and fan, gives

- Low-Pressure Compressor (*LPC*):

$$\dot{W}_{lpc} = \dot{m}_c C_{pc} (T_{t2.5} - T_{t2}) \tag{29}$$

- High-Pressure Compressor (*HPC*):

$$\dot{W}_{hpc} = \dot{m}_c C_{pc} (T_{t3} - T_{t2.5}) \tag{30}$$

- Low-Pressure Turbine (*LPT*):

$$\dot{W}_{lpt} = \dot{m}_6 C_{pt} \eta_{m-lpt} (T_{t6} - T_{t7}) \tag{31}$$

- High-Pressure Turbine (*HPT*):

$$\dot{W}_{hpt} = \dot{m}_4 C_{pb} \eta_{m-hpt} (T_{t4} - T_{t5}) \tag{32}$$

- Fan:

$$\dot{W}_{fan} = \dot{m}_{fan} C_{pc} (T_{t13} - T_{t2}) \tag{33}$$

For a dual-spool gas turbine engine, *HPT* and *HPC* are connected through a single shaft. Power extracted by *HPT* will be completely consumed by *HPC*, i.e.,

$$\dot{W}_{hpc} = \dot{W}_{hpt} \tag{34}$$

Accordingly, the total temperature ratio (τ_{hpt}) across the *HPT* becomes

$$\tau_{hpt} = 1 + \frac{1 - \tau_{hpc}}{(1 + f_b)\left(\dfrac{C_{pb}}{C_{pc}}\right) \tau_{hpc} \tau_b \eta_{m-hpt}} \tag{35}$$

where τ_{hpc} is one of the design parameters.

On the other hands, the *LPC*, *LPT*, and fan are connected through another shaft. For a turbofan, they are related by the following equation:

$$\dot{W}_{lpc} + \dot{W}_{fan} = \dot{W}_{lpt} \tag{36}$$

Similarly, the total temperature ratio (τ_{lpt}) across the *LPT* becomes

$$\tau_{lpt} = 1 + \frac{1 - \tau_{lpc} + \alpha(1 - \tau_{fan})}{(1 + f)\left(\dfrac{C_{pt}}{C_{pc}}\right)\left(\dfrac{T_{t6}}{T_{t2}}\right) \eta_{m-lpt}} \tag{37}$$

where τ_{lpc} is one of the design parameters and α is fan *BR*:

$$\alpha = \frac{\dot{m}_{fan}}{\dot{m}_c} \tag{38}$$

Although the powers are balanced individually ("two-unmixed-spool analysis"), power amounts between the *HPC* and *LPC* are not specified directly. Instead, the *HPC* and *LPC* total pressure ratios are split based on these two relations shown:

$$\pi_c = \pi_{lpc} \cdot \pi_{hpc} \tag{39}$$

$$\pi_{hpc} = A \cdot \pi_{lpc} \tag{40}$$

where *A* is a user input parameter, the value of which depends on the design. Based on the selected values of (π_{lpc}=3.2, π_{hpc}=12.5), 3.90 is therefore selected for *A* in this study. Furthermore, the relation between π_{lpc} and π_{hpc} is not limited to be linear.

Turbofan engine performance

As indicated, the *ST* is defined as the total uninstalled thrust (through core engine and fan) per unit mass flow rate intake,

$$ST = \left(\frac{F_c}{\dot{m}_c}\right) + \alpha \left(\frac{F_{fan}}{\dot{m}_{fan}}\right) \tag{41}$$

and *TSFC* is defined as the total fuel flow rate (main burner and *ITB*) per unit thrust,

$$TSFC = \frac{f_b + f_{itb}}{ST} \tag{42}$$

Rearranging equation (3), the *thermal efficiency* can be computed as

$$\eta_{th} = \frac{\dfrac{a_0^2}{2g_c}\left\{\left[(1 + f_b + f_{itb})\left(\dfrac{V_{10}}{a_0}\right)^2 - M_0^2\right] + \alpha\left[\left(\dfrac{V_{19}}{a_0}\right)^2 - M_0^2\right]\right\}}{f_b \cdot (h_{PR-b}) \cdot \eta_b + f_{itb} \cdot (h_{PR-itb}) \cdot \eta_{itb}} \tag{43}$$

The Computer Codes

An *Excel* program was written in combination among spreadsheet neuron cells, *Visual Basic*, and macro code to provide user-friendly interface so that compilation and preprocessing are not needed. A *Fortran* code was also written based on the theory just described. The results computed using both codes were found to be consistent, regardless of the code precision used. The input data (operating conditions) for a two-spool, separate-flow turbofan engine with and without *ITB* were provided by NASA Glenn Research Center.

Two configurations were used in this study, i.e. base turbofan engine without *ITB* and turbofan engine with *ITB*. When *ITB-OFF* option is chosen, the program will execute as if there is no *ITB* added to the turbofan engine. Accordingly, the following variables will be set internally and automatically:

- $C_{pt} = C_{pb}$, $\gamma_t = \gamma_b$, and gas properties will be the same downstream of the main burner
- $f_{itb} = 0$, i.e., no fuel injected into the *ITB*
- $\pi_{itb} = 1.0$ and $\tau_{itb} = 1.0$, i.e., no pressure drop and temperature change across station 5 and 6

IV. Results and Discussions

For a range of flight Mach number, four engine design choices, namely (1) *CPR*, (2) *FPR*, (3) fan *BR*, and (4) linear relation of *HPC* and *LPC* total pressure ratios, and one design limitation (*HPT* inlet temperature) are studied. These design parameters are then used to obtain the system performance parameters of *ST*, *TSFC*, and η_{th}. Other design parameters, such as compressor/turbine efficiency, combustor efficiency, and pressure drop/increase across various components, are user-defined input parameters, as shown in Table 1.

Flight Mach number

Figure 3 shows the performance comparison for the turbofan engines with and without *ITB* at flight Mach number in the range of 0.0 to 3.0. For each engine, the *CPR* and *FPR* are fixed at 40 and 1.65 respectively with maximum allowable *HPT* inlet temperature (T_{t4}) of 1500K and maximum *ITB* exit temperature (T_{t6}) of 1900K. Relative to base engines at both *BR*s (1 and 6), adding *ITB* will increase *ST* by 25% with only a small increase in *TSFC* as flight Mach number increases. In addition, *ITB* engine allows a wider range of flight operation than conventional base engines. These trends of performance qualitatively agree with the finding of Liu and Sirignano[1].

As shown in Figure 3, it is clear that a high-*BR* base engine can operate at supersonic speed with an addition of *ITB*, without any penalty of very high fuel consumption, as long as the flight Mach number is less than 2.3. However, the situation may deteriorate if the aerodynamic effect is considered. On the contrary, engine at low-*BR* does not have this problem because of the smaller frontal area.

Compressor pressure ratio

Figure 4 compares the engine performance for varying *CPR* at a supersonic speed (M_0=1.5) with T_{t4} = 1500K and T_{t6} = 1900K. The *FPR* is fixed at 1.65. As *CPR* increases, all engines at both *BR*s exhibit a decrease in *ST* and thermal efficiency with an increase in *TSFC*. For a conventional base engine, higher *CPR* limits the heat addition in the main burner due to the higher inlet temperature of the incoming air. The situation is even worse at the supersonic flight when the ram effect introduces at least a pressure rise of 2.0 times the ambient pressure, which raises further the inlet air temperature of the main burner. The consequence is the decreasing trend of thermal efficiency as shown in Figure 4c. *ITB* remedies this problem by allowing secondary heat addition at a pressure relatively higher than the pressure of an afterburner at some military engines. Nevertheless, *ITB* engine is superior to base engine because it gains more than 50% increase in *ST* with only less than 20% increase in *TSFC* at *CPR*

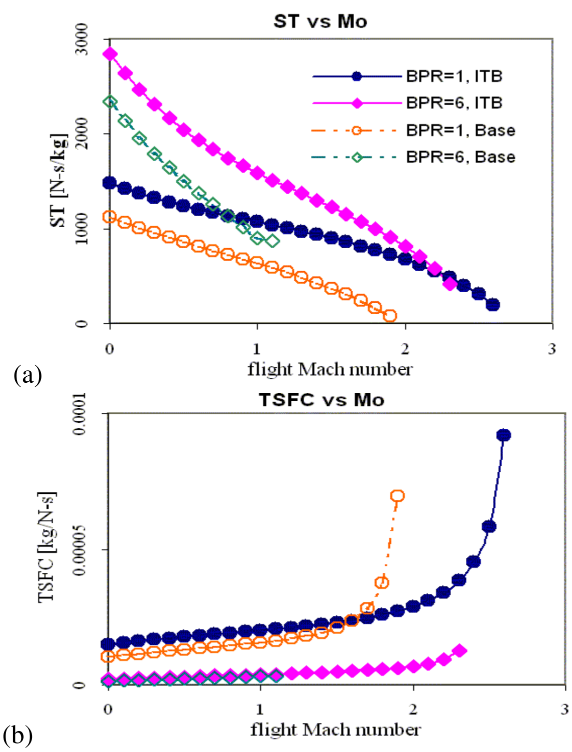


Figure 3 Performances of turbofan engines vs. flight Mach number at π_{lpc} = 3.2, π_{hpc} = 12.5, π_f = 1.65, T_{t4} = 1500K, and T_{t6} = 1900K.

greater than 20.

Holding all parameters constant, consider a typical military turbofan engine with a higher *FPR* of 4.0 (multi-stage fans) and a low *BR* of 1.0. The results are shown in Figure 5. One can observe the similar trend as discussed above, except that in Figure 5b, where the intersection point of the two engines' *TSFC* curves is shifted to a lower *CPR* value (about 32) when comparing with Figure 4b. Beyond that point, the *TSFC* of a conventional base engine will increase exponentially whereas *ITB* engine's *TSFC* stays steady as *CPR* increases. Clearly, *ITB* can be a potential improvement to the military supersonic turbofan engine performance.

Figure 6 shows the performance comparison for the same types of engines at M_0 = 0.87, which is at the subsonic flight. Because of the similar trend of performance, what we have observed and discussed in the supersonic flight conditions can be equally applied to the subsonic flight conditions. However, at subsonic flight, pressure rise due to ram effect is lower, i.e. about 1.6 times the ambient pressure (at M_0 = 1.5, pressure rise is 3.67). Therefore, for *ITB* engines at both *BR*s, when *CPR* is greater than 30, *ST* and *TSFC* are almost independent of *CPR*.

Fan pressure ratio

Increasing *FPR* is a way to supply more energy to bypass flow. In Saravanamuttoo et al.[5], military engines may have two- or three-stages fan with *FPR* as high as 4.0 whereas civil engines will always use a single-stage fan with *FPR* of about 1.5 to 1.8. Now, let us focus our

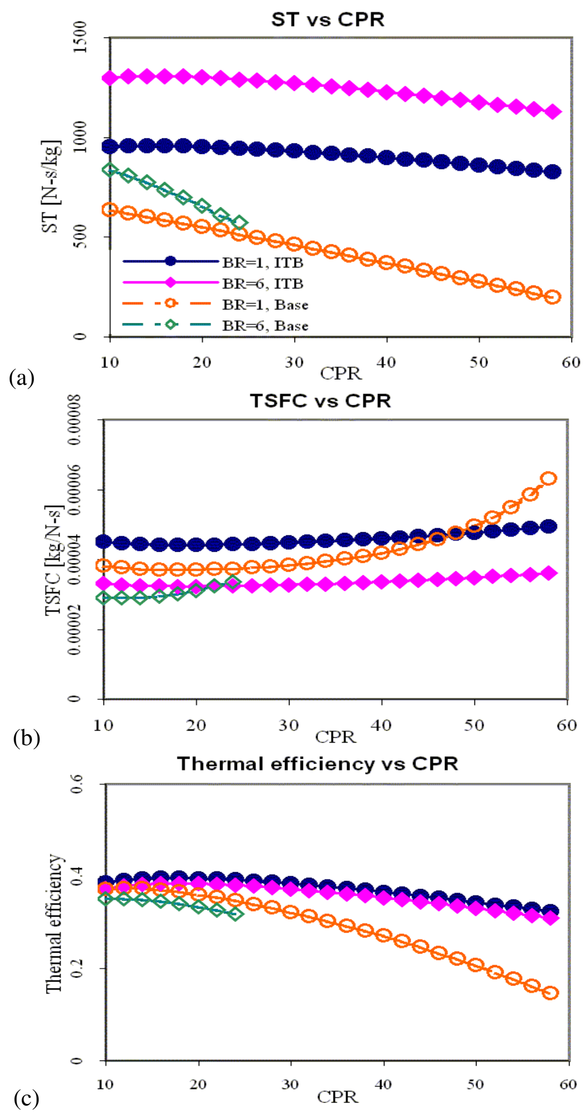


Figure 4 Performances of turbofan engines vs. *CPR* at M_0 = 1.5, π_f = 1.65, T_{t4} = 1500K, and T_{t6} = 1900K.

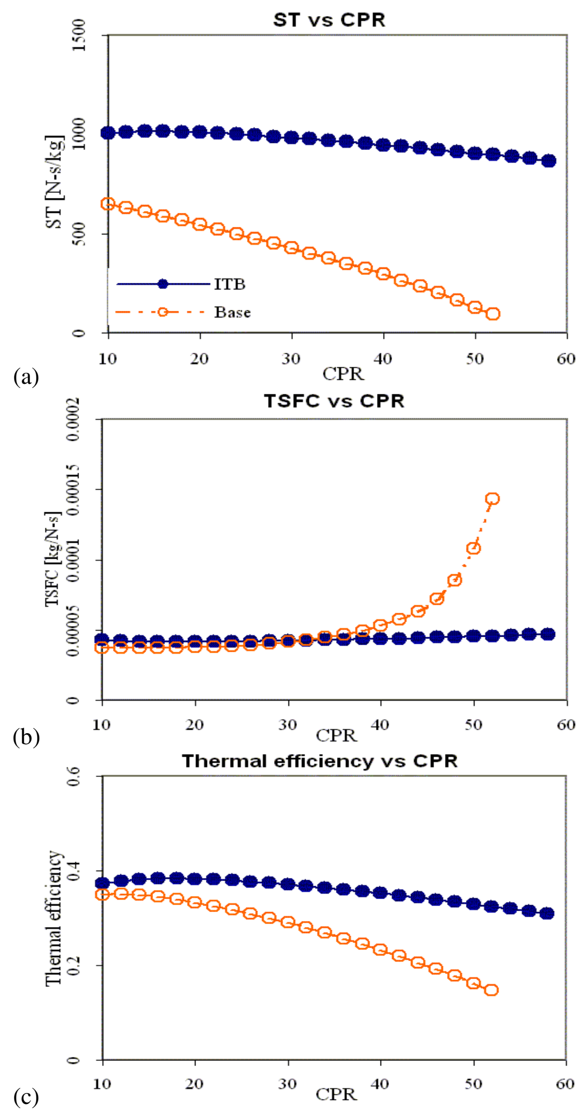


Figure 5 Performances of turbofan engines vs. *CPR* at M_0 = 1.5, π_f = 4.0, T_{t4} = 1500K, and T_{t6} = 1900K.

attention on the engines with *BR* = 1.0 where most supersonic military aircrafts use to keep the frontal area down. As shown in Figure 7, *ITB* engine gains the benefit of increasing *FPR*, where *ST* increases and *TSFC* decreases gradually. For base engine, *ST* increases initially and starts decreasing at *FPR* = 2.0, because more work is extracted from *LPT* to fan in order to achieve the specified *FPR*. Unfortunately, lower energy at *LPT* exhaust stream results in lower average exit velocity of the fan and the engine core stream, and thus lower thrust. However, the secondary heat addition in *ITB* supplies more energy to *LPT* to drive the fan with only slight increase in *TSFC*. Furthermore, as shown in Figure 7b, *TSFC* of *ITB* engine is becoming lower than that of the base engine at *FPR* beyond 2.7, indicating that *ITB* operates more efficiently.

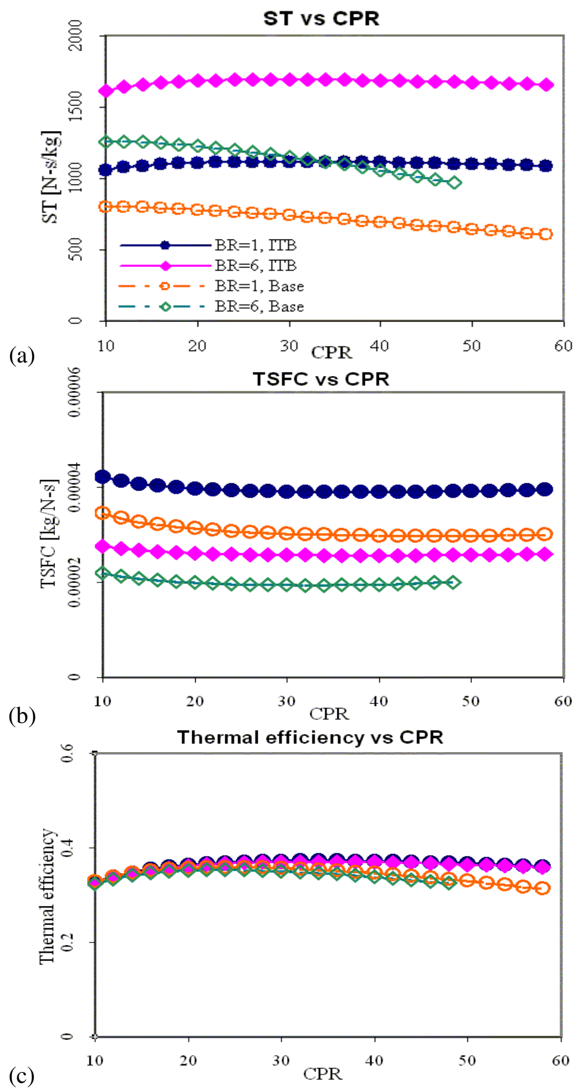


Figure 6 Performances of turbofan engines vs. *CPR* at M_0 = 0.87, π_f = 1.65, T_{t4} = 1500K, and T_{t6} = 1900K.

For subsonic flight, performance trend is qualitatively similar to the supersonic flight condition and will not be shown here.

Fan bypass ratio

Figure 8 shows the performance comparison for varying fan *BR* at two *FPR* settings. Clearly, as *BR* increases, base engine with high *FPR* (i.e., 4.0) exhibits an exponential-like increase in *TSFC* at a supersonic cruise. It ceases to produce thrust at *BR* beyond 1.3. However, adding *ITB* to this engine will not only widen its operation range up to moderate *BR* (say 3.0), but also gain more than 100% increase in *ST* accompanied by a decreasing trend of *TSFC* as *BR* increases. It may be unfeasible to operate a supersonic engine at a moderate *BR*. Nevertheless, advance in turbine technology (increased T_{t4}) will soon allow using a fan with *BR* larger than those traditionally used in supersonic turbofan engines (*BR* = 0.5 or less) while maintaining a reasonably small frontal area[6].

For a subsonic flight, Liu and Sirignano[1] clearly indicated the benefit of increasing fan *BR* on *ST*. Not surprisingly, our results (not shown here) show similar qualitative trend as in Liu and Sirignano[1]. However, as seen on some gas turbine literature, the inlet air mass flow, $\dot{m}_{air}$, in equation (1) is sometimes defined as the

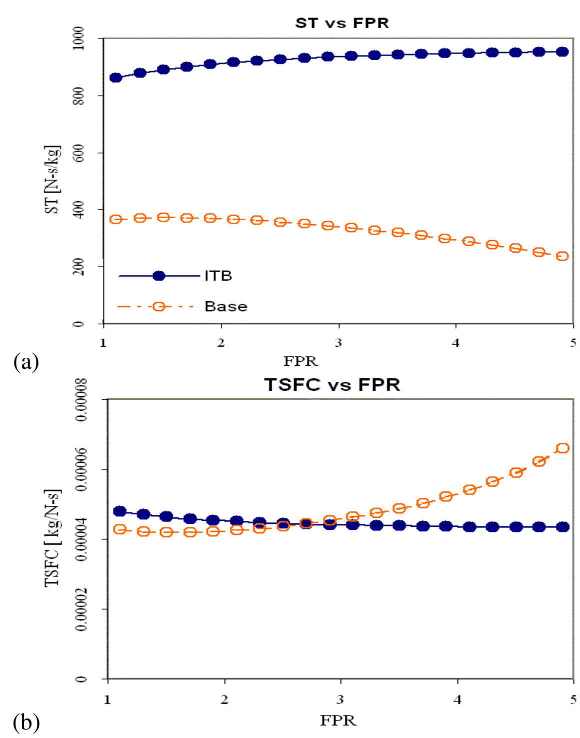


Figure 7 Performances of turbofan engines vs. *FPR* at M_0 = 1.5, α = 1.0, π_{lpc} = 3.2, π_{hpc} = 12.5, T_{t4} = 1500K, and T_{t6} = 1900K.

total air mass flow rate of both engine core and fan. Consequently, one will see a decreasing trend of *ST* as fan *BR* increases.

Linear relation of *HPC* and *LPC* total pressure ratios

Comparing to an engine with a parameter A = 1, A greater than 1 implies that 'more power' is needed to drive the *HPC*. On the other words, an increase in A results in more power produced at *HPT* or reduction of the inlet pressure and temperature in *ITB*. From Figure 9, one can see that both *ST* and *TSFC* are proportional to A. This is because more fuel can be burned in *ITB* in order to meet the specified *ITB* exit temperature requirement. However, an increase in *TSFC* implies that the rate increase in the amount of fuel injected is greater than the engine thrust produced. For a given large A value, one must notice that fuel is now burned at a much lower pressure. Possibly, the *ITB* engine is operating like an afterburner engine for a very large A. In addition, as A increases, thermal efficiency decreases faster than the engine configuration with lower A. It is clear that, for a given turbofan engine configuration, parameter A is an important engine design variable. One wants to maximize and optimize A to have a more control over engine thrust.

Variation of A value between 0 and 1 constrains the amount of heat addition in the *ITB* because of smaller work required to drive *HPC*, or smaller work produced by *HPT*. As a result, there is only a small amount of temperature drop across *HPT*, and thus higher inlet temperature of *ITB*. Accordingly, restricting A value between 0 and 1 is not desirable.

***HPT* Inlet Temperature**

For a base engine with *BR* = 1.0 and *FPR* = 4.0, the minimum T_{t4} has to be at least 1400K as shown in Figure 10a. With *ITB*, T_{t4} can be as low as about 1100K while producing more *ST* and reducing *TSFC*. Further increase of *BR* to a moderate value, say 3.0, holding other parameters the same, base engine will not produce any thrust at all. However, as shown in Figure 10a, a *BR*=3.0 powered *ITB* engine at T_{t6} = 1600K (not 1900K as used in all other cases) is still able to operate at T_{t4} greater than 1500K. Preliminary result (not shown here) indicates that, at higher T_{t6}, the same engine can operate at lower T_{t4}, i.e. 1400K, yet producing more *ST* with less *TSFC*. From preceding discussion, it is clear that the addition of *ITB* makes the operation of a supersonic turbofan engine up to a moderate *BR* (=3.0) possible. However, as shown in Papamoschou[6], in order to keep the frontal area small, it requires advanced turbine technology for a very high T_{t4}. It may be interesting to investigate whether one can apply the *ITB*'s advantage (i.e., lowering T_{t4} through addition of *ITB*) to keep the same T_{t4} while maintaining a small frontal area.

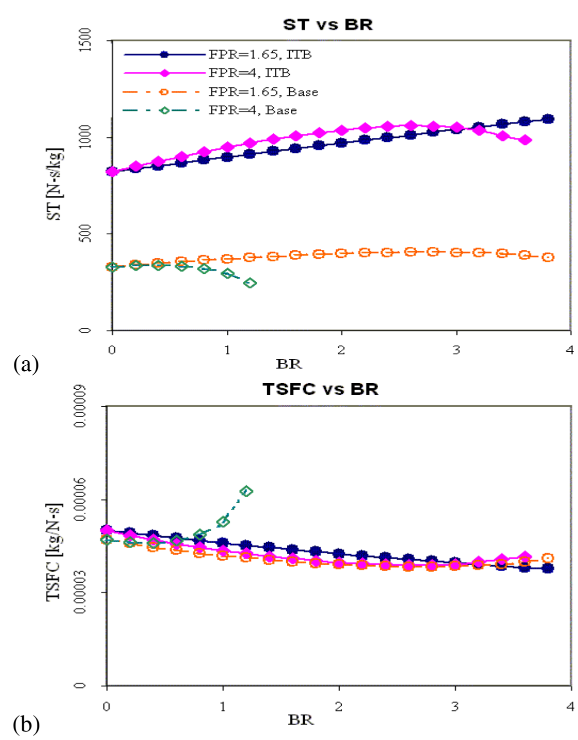


Figure 8 Performances of turbofan engines vs. fan *BR* at M_0 = 1.5, π_{lpc} = 3.2, π_{hpc} = 12.5, T_{t4} = 1500K, and T_{t6} = 1900K.

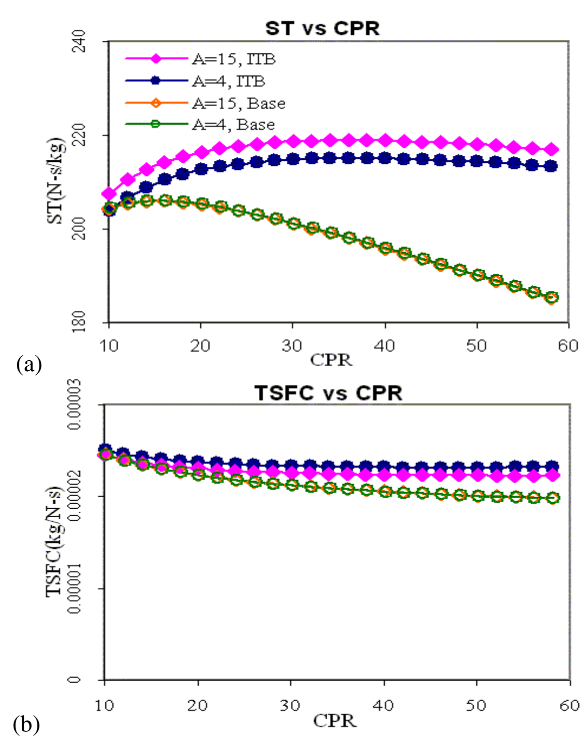
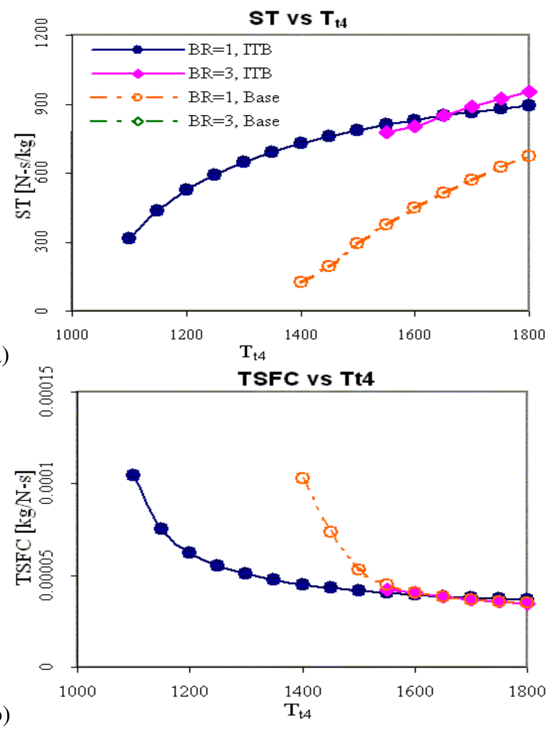


Figure 9 Performances of turbofan engines vs. *CPR* at different values of A, M_0 = 0.87, π_{lpc} = 3.2, π_{hpc} = 12.5, π_f = 1.65, T_{t4} = 1500K, and T_{t6} = 1900K.

For subsonic flight, *ITB* engines benefit equally well from lower T_{t4} as discussed in supersonic flight.



(a)

(b)

Figure 10 Performances of turbofan engines vs. T_{t4} at M_0 = 1.5, π_{lpc} = 3.2, π_{hpc} = 12.5, π_f = 4.0, T_{t4} = 1500K, and T_{t6} = 1600K.

V. Conclusions

Results of the parametric studies presented in this paper can be summarized as followings:

1) An *ITB* engine gains more benefit when operating at high flight Mach number. It provides a design basis for high-performance engines applicable to lightweight and/or high-speed aircraft.

2) Allowing heat addition in the *ITB* without a significant increase of *TSFC* further extends the operational range of compressor pressure ratio. At subsonic flights, *ITB* engine maintains almost the same level of *ST* and *TSFC* over the operating range.

3) Most low-*BR* supersonic engine with high *FPR* benefits from *ITB*, because it does not have the penalty of very high fuel consumption, yet producing higher *ST*.

4) With *ITB*, supersonic engine now can operate at a moderate *BR*.

5) Value for the linear relation between *HPC* and *LPC* total pressure ratios, i.e., parameter *A*, must be greater than 1 and its value requires optimization for a given mission.

6) Through the addition of *ITB*, *HPT* inlet temperature can be lower while producing more *ST* with less *TSFC*.

Although there are many advantages of using *ITB* for better engine performance, there are also challenges needed to be resolved. Specific hardware design challenges are:

- High velocities with possible swirl in the transition duct are important issues regarding the flame stability.
- Higher *LPT* temperatures than the conventional engine will require redesign of the *LPT* cooling system with a reduction in *LPT* stages.
- Integration and complexity of a second combustor, including all associated cooling and control requirements, needs to be overcome.

The current computer program is written for a specific engine configuration, namely unmixed two-spool turbofan engine with a separate fan and engine core stream nozzles. Application of an *ITB* is not limited to this configuration. Therefore, to increase the flexibility

Description	Input value
Polytropic efficiency	
Fan (e_{fan})	0.8961
Low-pressure compressor (e_{lpc})	0.9036
High-pressure compressor (e_{hpc})	0.9066
High-pressure turbine (e_{hpt})	0.9029
Low-pressure turbine (e_{lpt})	0.9174
Total pressure ratio	
Inlet ($\pi_{d,max}$)	0.99
Main burner (π_b)	0.96
ITB (π_{ITB})	0.96
Nozzle (π_n)	0.99
Fan nozzle (π_{fn})	1.3
Component efficiency	
Main burner (η_b)	0.99
ITB (η_{ITB})	0.99
Mechanical	
Low-pressure spool (η_{m-lp})	0.93
High-pressure spool (η_{m-hp})	0.92
Specific heat at constant ratio (kJ/kg-K)	
Region* 0→3 (C_{pc})	1.004
Region 4→5 (C_{pb})	1.096
Region 6→10 (C_{pt})	1.089
Specific heat ratio	
Region 0→3 (γ_{pc})	1.399
Region 4→5 (γ_{pb})	1.273
Region 6→10 (γ_{pt})	1.279
Fuel low heating value (h_{PR})	42798.4 kJ/kg

Table 1 – input parameters
*Region numbers refer to Figure 2.

and usefulness of the program developed, additional options for different engine configurations would be a very desirable feature. Currently, a research on the performance (*off-design*) cycle analysis of a turbofan engine with an *ITB* is on-going.

Nomenclature

A	cross-sectional area or linear constant
a	sound speed
C_P	specific heat at constant pressure
e	polytropic efficiency
F	uninstalled thrust
f	fuel/air ratio
g_c	Newton's constant
h_{PR}	low heating value of fuel
M	Mach number
$\dot{m}$	mass flow rate
P	pressure
P_t	total pressure
R	universal gas constant
T	temperature or installed thrust
T_t	total temperature
V	absolute velocity
$\dot{W}$	power

Greek symbols

α	bypass ratio
γ	ratio of specific heats, c_p/c_v
η_m	mechanical Efficiency
η_{th}	thermal Efficiency
π	ratio of total pressure
π_r	(exception) ratio between total pressure and static pressure due to the ram effect, P_t/P
τ	ratio of total temperature
τ_r	(exception) ratio between total temperature and static temperature due to the ram effect, T_t/T
τ_λ	ratio between total enthalpy and enthalpy at ambient condition

Subscripts

$_b$	main burner or properties between main burner exit and ITB
$_c$	properties between upstream and main burner or engine core
$_d$	diffuser
$_f$	fuel
$_{fan}$	fan
$_{fn}$	fan-nozzle
$_{hpc}$	high pressure compressor
$_{hpt}$	high pressure turbine
$_{itb}$	interstage turbine combustors
$_{lpc}$	low pressure compressor
$_{lpt}$	low pressure turbine
$_0$	inlet
$_n$	nozzle
$_r$	ram
$_t$	properties between ITB exit and downstream or total/stagnation values of properties (i.e. temperature, pressure or enthalpy)

Abbreviations

BR	Bypass Ratio
CPR	Compressor Pressure Ratio
FPR	Fan Pressure Ratio
ST	Specific Thrust
TSFC	Thrust Specific Fuel Consumption
LPC	Low-pressure Compressor
HPC	High-pressure Compressor
ITB	Interstage Turbine Burner

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13. ABSTRACT ***(Maximum 200 words)***
Today's modern aircraft is based on air-breathing jet propulsion systems, which use moving fluids as substances to transform energy carried by the fluids into power. Throughout aero-vehicle evolution, improvements have been made to the engine efficiency and pollutants reduction. This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (ITB). The ITB considered in this paper is a relatively new concept in modern jet engine propulsion. The JTB serves as a secondary combustor and is located between the high- and the low-pressure turbine, i.e., the transition duct. The objective of this study is to use design parameters, such as flight Mach number, compressor pressure ratio, fan pressure ratio, fan bypass ratio, linear relation between high- and low-pressure turbines, and high-pressure turbine inlet temperature to obtain engine performance parameters, such as specific thrust and thrust specific fuel consumption. Results of this study can provide guidance in identifying the performance characteristics of various engine components, which can then be used to develop, analyze, integrate, and optimize the system performance of turbofan engines with an ITB.

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